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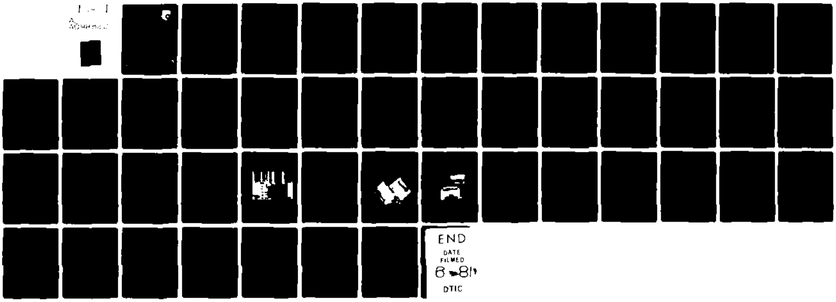

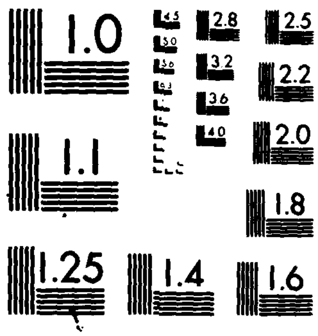

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THE INFLUENCE OF HEAT TRANSFER ON THE DRAG OF AIRFOILS

DR. JOHN D. LEE

The Aeronautical and Astronautical Research Laboratory
The Ohio State University
Columbus, OH 43212

APRIL 1981

Final Report May 1979 to September 1980

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Robert A. Large

ROBERT A. LARGE, Capt, USAF
Project Engineer

Lowell C. Keel

LOWELL C. KEEL, Maj, USAF
Chief, Aerodynamics & Airframe Branch
Aeromechanics Division

FOR THE COMMANDER

PETER J. BUTKEWICZ, Colonel, USAF
Chief, Aeromechanics Division

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An experimental and theoretical study was performed to determine the effects of surface temperature on the drag of airfoils. Models of an aft-loaded profile and of a NACA 65A413 were tested with separate models providing pressure distributions and drag data from cooling or heating at a Mach number of 0.4 and Reynolds number of 2.5×10^6. Simultaneously an airfoil boundary layer analysis was performed using available theories and empirical correlations for laminar, transitional and turbulent boundary layers.

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Cooling was found to provide a lower drag and heating a higher value, as expected, for both profiles and the reduction by cooling was substantial for the aft-loaded profile at the condition tested. The measurements agreed well with the theoretical predictions which may then be used to evaluate the effects of both heat transfer and pressure gradients on airfoil transition and drag.

FOREWORD

This is the final report on an experimental and theoretical study to determine the effects of surface temperature on the drag of airfoils. This work was performed by the Aeronautical and Astronautical Research Laboratory of the Ohio State University under contract number F33615-79-C-3008 for the Flight Dynamics Laboratory.

This work was funded by the Flight Dynamics Laboratory Director's Fund. It was performed under work unit 24041037. The principle investigator on this effort was Dr. John D. Lee, The Ohio State University. Technical direction was provided by Capt Robert A. Large of the Air Force Wright Aeronautical Laboratories/FIMM.

This report is also identified as RF711985 for Ohio State's internal control purposes.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF FIGURES (continued)

LIST OF SYMBOLS

c	airfoil chord
C_d	drag coefficient
C_ℓ	lift coefficient
C_p	Pressure coefficient
F	factor for turbulence in transition equation
H	ratio of displacement to momentum thicknesses
K	Cohen-Reshotko gradient parameter
L_n	Natural logarithm
M	Mach number
M_1	Mach number outside the boundary layer
R_e	Reynolds number
T_f	film temperature of boundary layer
T_w	surface temperature of airfoil
T_o	stagnation temperature of free stream
T_1	static temperature outside the boundary layer
t	trailing edge thickness
U	local free stream velocity
X	chordwise or surface distance
α	angle of attack
δ^*	displacement thickness
θ	momentum thickness
ν	kinematic viscosity
ρ	density
τ	shear stress

INTRODUCTION

The possibility of a considerable drag reduction by using the fuel of a cryo-fueled aircraft to stabilize the laminar boundary layers before burning it has been examined by Reshotko[1]. The feasibility rests on an accurate engineering prediction of the combined effects of heat transfer and pressure gradient on transition and the experimental verification of the prediction technique under conditions typical of flight. The influences of compressibility, heat transfer and pressure gradient have been studied extensively for both laminar and turbulent boundary layers as in, for example, Cohen and Reshotko[2] and van Driest[3] for which there is a considerable body of experimental verification. Airfoil flow fields may be predicted with acceptable accuracy for moderate conditions for both subcritical[4,5] and supercritical[6] flows and the corresponding computer codes use boundary layer prediction techniques as developed in the referenced works.

However, for the interfacing problem of transition from laminar to turbulent flow there are few useful theoretical analyses and available experimental data[7,8,9] are generally applicable to much higher speeds and more extreme conditions than the current interest. Boehman and Mariscalco[10] made a valuable summary of the state-of-the-art to 1976 and included a transition-predicting technique applicable in some cases to airfoils in the sub- and transonic speed range[11].

The prediction of transition is a key item in determining the drag of an airfoil with variations in surface temperature.

Compressible boundary layer theory demonstrates that cooling the surface causes an increase in skin friction, whether laminar or turbulent, due to a reduction in thickness. However, the drag may be reduced by a rearward shift in transition, replacing turbulent flow by laminar, with its much lower skin friction. A proper analysis must then include the determination of transition-shifting as well as the thermal changes in the boundary layers. In addition, separation must be correctly predicted so that the criteria for separation must be re-evaluated.

The present study was undertaken to provide experimental evidence on the thermal effects on transition for airfoils at Mach numbers, Reynolds numbers and lift coefficients typical of flight. At the same time theoretical predictions were to be made using the available airfoil codes and the corresponding transition criteria were to be modified or replaced.

The experiments were undertaken using brass airfoil models cooled by liquid nitrogen or heated by hot water, in the OSU 6 inch by 22 inch airfoil tunnel. This wind tunnel has a documented, low level of interference[12] and has been used for a wide variety of research and development studies for both governmental agencies and members of the aerospace industry. The theoretical predictions were undertaken in the Laboratory's Airfoil Design and Analysis Center (ADAC), originally sponsored by NASA[13] (Langley Research Center).

The research study centered on two profiles: one an aft-loaded ("supercritical") section, the other the NACA 65A413 representative of the NACA "laminar-flow" type. Pressure-tapped models were used to obtain surface pressure distributions for comparison with the theoretical code predictions and/or the direct inputs in analysing the boundary layer. The thermal models were not tapped in order to keep the surface free from any disturbances which might prematurely initiate transition, a feature which had been observed. The only datum taken relative to the thermal models, other than the model temperatures, was the wake pitot-deficit profile, by which drag was computed. Originally the intent was to obtain the wake temperature profile as well but an analysis showed that this was unnecessary for the temperature differences available in the test program.

Detailed examination of the boundary layer subroutines used in the airfoil analysis code showed that they were inadequate for this study and that useful revisions were not possible. A replacement code was developed using more advanced methods for handling compressibility and heat transfer and a compatible, thermally oriented transition-predicting method. Data are presented comparing the experimental measurements with the theoretical predictions.

EXPERIMENTAL PROGRAM

A. Test Facility

The experimental phase of the program was conducted in the OSU 6 inch by 22 inch airfoil tunnel. Reference 12 contains a detailed description of the wind tunnel and the associated systems; the following points are pertinent to this study. The test section is 6 inches wide by 22 inches high and 44 inches long (streamwise, from the nozzle exit to the collector). It has a "two-dimensional" configuration, the sidewalls being solid, plane and parallel while the top and bottom walls are perforated, each backed by a two-inch deep chamber to accommodate transonic flow. The fact that the two backup chambers are separated from each other results in a low level of confinement interference. Mach number in the test section is adjusted by an array of "choke-bars" downstream of the test section.

Figure 1 shows the test section and the method of mounting models in it together with the scanivalve system used for acquiring pressure data. The model port may be rotated to change the attack angle. Not visible in the photo is a motor-driven pitot probe which traverses the wake of a model, approximately 6 inches downstream from the trailing edge in the center of the stream.

The wind tunnel was calibrated by measurements of static pressure taken at the model port location (tunnel empty) and referenced to the plenum chamber pressures (averaged). The interference was calibrated by comparing pressure distributions on models differing in chord by a factor of two. A symmetric profile (NACA 0012) and an aft loaded profile were used to determine that the interference was essentially a downwash of 0.17° per C_L.

The particular nozzle and test section configuration has resulted in a highly uniform flow field without the use of any treatment of the sidewall boundary layers. Extensive measurements have shown that the spanwise pressure distributions on several types of airfoil models do not exhibit variations in C_p from centerline to the sidewall of more than 0.02, short of stall.

Static pressure data are acquired through scanivalve-stop-valve sets. Pressurized tubes store samples from the model taps for post-run measurements through the scanivalves. Since the volume at the transducer is finite, though small, the subsequent readout must be adjusted according to the results of previous calibrations.

Data acquired as above is stored on tape in the Digital Computational Facility at AARL for post-run reduction. The results are available for review on a CRT in about two minutes after a run in coefficient form, with hard copy tabulations and plots available at that time also.

B. Models

Profiles of the test models are presented in Figure 2. Pressure models (as shown in Figure 3) having a chord of 6 inches were used to obtain the experimental pressure distributions for comparing with the predictions from the airfoil codes and for direct inputs into the boundary layer theory. The thermal models (Figure 4) had chords of 4 inches and contained the coolant passages as well as a single thermocouple to monitor the temperature, located at midspan at about the 30 percent chord point. Otherwise the thermal models had no instrumentation to mar the surface and possibly initiate transition.

All the models were machined from solid blocks of brass on a numerically-controlled milling machine; a minimum of 1200 coordinate pairs were used to describe the surfaces. Coolant passages in the thermal models and tap holes in the pressure models were drilled after the surface was completed according to prescribed coordinates. The surfaces of the models were hand polished to a mirror finish before each series of tests. Coolant manifolds were attached to the sides of the thermal models outside the tunnel during testing.

C. Test Procedure

The usual operation of the wind tunnel is for a run period of about 15 to 20 seconds. The control valves are preset to deliver a selected pressure at the tunnel and upon opening the main valve the tunnel is pressurized in about 5 seconds. With a pressure model installed the stop valves are then closed, the wake surveyed and the tunnel shut down. The run time is thus dependent primarily upon the extent of the wake to be surveyed, being greater for stalled flow or supercritical flow than for attached, subcritical flow. Figure 5 shows typical wake profiles. Wake data are digitized and recorded during the survey while the scanivalves are operated post run to measure the trapped samples from the pressure ports. Reduction, display and printout follow, as described before.

With a thermal model installed in the tunnel, the run period was usually extended until the model temperature was observed to be at a desired value. The wake was then surveyed and the tunnel shut down. A number of different procedures was tried for the thermal models, both to shorten the stabilization time (and hence the run time) and to avoid model surface contamination. In all cases the coolant fluid was pressure-pumped from a large thermos bottle to the manifold on the model on one side of the tunnel, through the model to the manifold on the other side and from there to another thermos bottle.

The most successful procedure was to simultaneously start the coolant flow and the tunnel air flow, adjusting each to obtain the preselected values before probing the wake. This avoided the condensation of water vapor from the atmosphere (the tunnel exhaust being open to the atmosphere) and the problems with shedding model shields without damaging the wake probe or blocking the choke.

During the period in which the experiments were conducted the Laboratory's air supply system was undergoing repairs. This not only restricted the air supply available for testing but also the dew point was only about -40°F. Some care had to be taken in cooling the model with liquid nitrogen to not overshoot the dew point since, in such cases, a frost film formed on the model surface and early boundary layer transition was evident in the increased drags obtained. For these reasons the program was supplemented with tests in which the models were heated with hot water, in order to provide a useful range of temperature variation to evaluate the effects of heat transfer on drag.

THEORETICAL ANALYSIS

In a detailed examination of the airfoil codes used in the OSU-ADAC, the two codes used primarily for subcritical-flow and another code used primarily for supercritical flow were all found to be deficient for application to compressible boundary layer analyses. The supercritical code did not contain any laminar flow analysis but depended upon a user-specification of the transition point and then proceeded with a turbulent boundary layer (including iteration for displacement). Only one subcritical code contained compressible laminar, transitional and turbulent analyses with several iterations on both the external field and the viscous flow with an apparent inclusion of surface heat transfer effects. However the laminar flow analysis, a derivative from the Cohen-Reshotko theory[2], was accurate only for wall-to-stagnation temperature ratios from about 0.85 to 1.15; further, the transition criterion was found to produce a shift due to heat transfer in the opposite direction to that which stability theory and observations would predict. The necessary modifications to the codes were concluded to be so extensive as to be not feasible and that an independent compressible boundary layer analysis should be required for the thermal effects on airfoil drag.

As noted, the theoretical determination of the magnitude of the drag change by cooling an airfoil surface depends upon correctly calculating the laminar and turbulent boundary layers as well as the transition point. With transition fixed (either by natural or artificial means) cooling gives an increase in drag due to the thinner boundary layer. A decrease in drag may be affected only by moving the transition point downstream sufficiently far as to overcome the increase in friction by replacing turbulent flow with laminar. Thus the boundary layer analysis must include an accurate prediction of both the transition point location and the local friction coefficient.

The boundary layer code which was formulated consisted basically of three parts: a laminar flow analysis using the approach and the results of Cohen and Reshotko (hereafter referenced as "C-R"), a transition criterion based upon stability theory and a turbulent boundary layer analysis using the correlations of Squire-Young[14] and of Garner[15] but with modifications for heat transfer by the film-temperature approach[16].

In the laminar analysis, the necessary functions for a C-R analysis were analytically fitted for fixed temperature ratio and then locally interpolated from one temperature ratio to another. For example, the skin friction function was matched by a form

$$L = A + B \exp(CK)$$

where A, B and C are parameters determined for each temperature ratio of Reference 2 (K is the pressure gradient parameter). For any arbitrary temperature ratio, the parameters were then linearly interpolated between the given values. Thus, the C-R functions were continuously available over the entire map of the pressure gradient parameter and of the wall-to-stagnation temperature ratio from 0 to 2. An allowance was made by means of tangent linear extrapolation for excursions beyond the "normal" range of the C-R parameter. In application, the analysis starts with the stagnation point boundary layer and procedes downstream iterating twice over each interval for the C-R form of the momentum integral and testing at each increment for separation and for transition. Depending on the rate-change of the pressure gradient parameter, laminar separation is characterized as a bubble-transition or a complete separation.

The transition criterion was established as a combination of the results from several investigators as reported by Schlichting[17]. Exponential curves were matched to the calculations from stability theories for the influences of pressure gradient and temperature ratio; comparisons are given in Figures 6 and 7. However, the amplification to transition was replaced by a factor, F, in the stability equation. The resulting equation is, for the displacement thickness at transition,

$$\delta^* = F \cdot 645 \cdot \mathrm{Exp}(42.83K + 8.84(1 - T_W/T_O)) \cdot \nu / U$$

where K is the C-R pressure gradient parameter and recovery temperature is assumed equal to free-stream stagnation temperature. F is a factor determined from experiments; a value of 2.5 was found suitable for normally smooth flow whereas a value of F = 1 denotes transition immediately at the instability point. The argument for these simplifications is as follows:
(1) the influence of pressure gradient is so large that transition should be expected at small pressure gradients (2) the effects of small to moderate heat transfer are correctly described for small pressure gradient and (3) surface roughness

and stream turbulence may be readily simulated by the factor F.

No account was made for the zonal nature of transition; rather it was assumed to occur at a point and the turbulent boundary layer assumed as fully developed, with an initial momentum thickness equal to the laminar value at transition and with the shape factor ratio of the turbulent "flat plate" value, modified by pressure gradient according to Garner's equation (see below).

The turbulent boundary layer analysis used the Squire-Young law[14] for shear stress (evaluated locally).

$$\frac{\tau}{\rho U^2} = \frac{0.1527}{\left[L_n(4.075 \frac{U\theta}{\nu})\right]^2}$$

and Garner's empirical relation for the shape factor[15],

$$\left\{\frac{U\theta}{\nu}\right\}^{1/6} \theta \frac{dH}{dX} = e^{5(H-1.4)}\left[0.0135(1.4-H)-\left\{\frac{U\theta}{\nu}\right\}^{1/6} \frac{\theta}{U}\frac{dU}{dX}\right]$$

In application, however, these relations were modified by the film temperature concept[16]:

$$\frac{T_f}{T_1} = 1 + 0.515 \left\{\frac{T_w}{T_1} - 1\right\} + 0.0335\ M_1^{\ 2}$$

Where the surface is implied, the values of density and viscosity in the shear stress and H formulas were evaluated at the temperature T_f. The results so modified were used in the compressible form of the momentum integral,

$$\frac{\tau}{\rho U^2} = \frac{d\theta}{dX} + \frac{\theta}{U}\frac{dU}{dX}(2 + H - M_1^{\ 2})$$

to calculate the development of the turbulent boundary layer over the surface from transition to the trailing edge. A value of $H = 2.2$ was specified as the index for separation.

The momentum thickness was then transferred from the trailing edge (or the separation point, if separation occurred) to the free stream conditions by an iterative integration of the momentum equation. This procedure was selected over the empirical transfer techniques of Squire-Young or Goradia and Lilley[18] as being more accurate and of greater application. The drag coefficient is then

$$C_d = \frac{2}{C} (\theta_u + \theta_\ell)$$

where θ_u and θ_ℓ are the transferred momentum thicknesses for the upper and lower surfaces.

The additive drag for a blunt trailing edge was estimated as

$$\Delta C_d = 34 \ (t/2C)^{4/3}/C_d$$

which was deduced from the correlations reported in Hoerner[19]. t is the trailing-edge thickness and C_D is the drag before correction.

In practice, the boundary layer code has been interfaced with an existing airfoil analysis code (previously developed by the author). The analysis is a modified Theodorsen-transformation. No provision was made at this time to iterate i.e. to recalculate the outer field with the addition of the boundary layer displacement to the airfoil. However, the program has the provision to introduce the experimental pressure distribution for the input to the boundary layer routine; this was the procedure used for this study.

Results of the boundary layer calculations are reported in the next section.

RESULTS AND DISCUSSION

Both profiles were analysed by means of the available airfoil codes in subcritical flow to determine suitable attack angles and Reynolds numbers for demonstrating the influences of heat transfer. The primary criterion was to maintain very small or favorable pressure gradients for as great a distance as possible from the leading edge, on both upper and lower surfaces so that there would be little or no tendency for initiating transition by adverse pressure gradients. The pressure models were then tested at the selected conditions and the attack angle changed slightly to obtain the best practical condition. It was assumed that the machined profiles of the pressure and thermal models were duplicated as well as the test conditions for the two models for all practical purposes.

For example, Figure 8 shows pressure distributions on the aft loaded airfoil at two angles-of-attack. At the lower angle, the pressure gradients are favorable over most of the upper surface but the lower surface exhibits an early adverse gradient and hence early transition. At the slightly higher attack-angle the situation is reversed for the two surfaces. From such data, it was concluded that the optimum attack angle for this airfoil was between these two. Wind tunnel tests were performed with the pressure model to determine such an optimum.

The pressure distributions for the condition selected for testing the aft-loaded FDLA profile is shown in Figure 9. The initial favorable pressure gradients and the lack of any appreciable gradient over large portions of both the upper and lower surfaces makes this case almost ideal for testing the influence of heat transfer on an airfoil having a realistic lift coefficient.

Results from tests of the thermal model are presented in Figure 10 and compared with the predictions from the boundary layer analysis at that Reynolds number. The reason for the marked variation in drag is clear from Figure 11 which shows the theoretical predictions of the shifts in the transitions for both surfaces. Although the temperature change was not large between the extremes of cooling and heating, it was sufficient to cause the shifts indicated. Comparisons with the pressure distributions of Figure 9 resulted in the conclusion that those shifts were about the maximum that could be produced within the bounds allowed by pressure gradients.

The pressure distribution for the case selected for the testing of the model of the NACA 65A413 profile is given in Figure 12. The corresponding results from tests of the thermal model are compared with the theoretical predictions of drag in Figure 13. The reason for the smaller change in drag over the temperature range of the tests is clear from Figure 14 in the smaller shifts in transitions. Again comparisons with the pressure distribution would lead to the conclusion that, for this profile, pressure gradients dominated the boundary layer transition at that Reynolds number, thus making the thermal influence on drag almost insignificant.

The procedure was repeated for the two profiles in supercritical flow, at a free stream Mach number of 0.8. The pressure distributions are presented in Figures 15 and 16. However, tests with the thermal models as well as the theoretical analyses showed no important changes in drag due to surface temperature variations. The transition of the boundary layer was in all cases dominated by pressure gradient.

A significant series of tests was conducted with the aft-loaded FDLA profile at the conditions represented by the pressure distributions of Figure 15 and a adiabatic wall temperature. The theoretical analysis predicted that the boundary layer on the upper surface was laminar and stable to the shock site where it separated completely. This behavior was verified by the results shown in Figure 17 in which the affects of a trip at 0.6C are indicative of forced transition and the retention of a turbulent boundary layer with a resulting lowered drag.

CONCLUSIONS

The speculation that cooling the surfaces of an aircraft wing will reduce the drag has been confirmed in this study, with some reservation. Experiments were conducted with models of two different airfoil profiles and drags were calculated using a thermally dependent transition criterion, with reasonably good agreement.

The two sections selected for the test program were an aft-loaded "supercritical" profile and the NACA 65A413, a conventional "laminar-flow" profile. The available airfoil analysis codes were used to determine pressure distributions which might be suitable and pressure-tapped models were tested in the OSU 6 in. by 22 in. airfoil tunnel to finalize the selection of the flow conditions. Drags were measured for thermal models, having untapped surfaces, cooled by liquid nitrogen or heated by hot water to provide a moderate range of surface temperature below and above adiabatic.

A boundary layer analysis code was developed to supplement those in the existing airfoil analysis codes which were inadequate for a realistic thermal analysis. The procedure used analyses of both the laminar and turbulent boundary layers which include the influences of pressure gradient, heat transfer and compressibility over ranges more than adequate for sub- and transonic airfoil flows and containing a corresponding compatible transition criterion.

Transition was noted to be primarily dependent upon pressure gradient in the range of the test program. However, in the case of the aft-loaded profile a considerable drag reduction was found for a moderate temperature change, at a realistic lifting condition due to a considerable shift in transition by cooling. The same profile at a supercritical flow condition exhibited severe separation effects following a long run of stable laminar flow, which was relieved by the addition of a trip to affect transition.

The boundary layer analysis technique appears suitable for making realistic engineering predictions and the experiments have verified that, under suitable conditions, transition may be thermally shifted to reduce overall drag. The procedure may be applied by combining the analysis with modern airfoil design techniques. The result would be the development of airfoil profiles or families of profiles for the exploitation of the control of transitions by natural means, i.e. by both heat transfer and pressure gradient, including the addition of forced transition to terminate overly-stable laminar flows. In suitable combinations, comparatively large reductions in drag should be attainable.

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Figure 1. Photograph of Test Section of 6 in. by 22 in. Wind Tunnel Showing Model Part and Scanivalve Arrangement

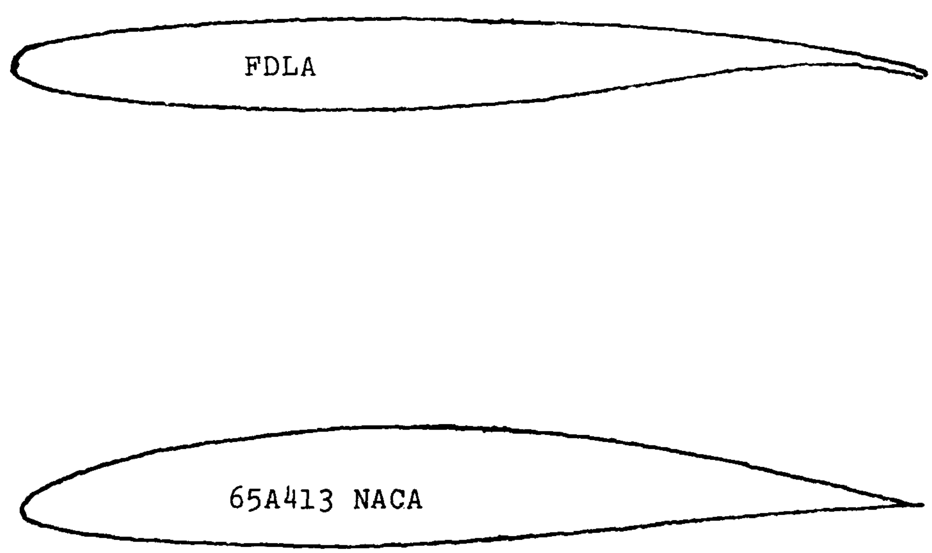


Figure 2. Outlines of the Two Airfoil Profiles Used in the Current Study.

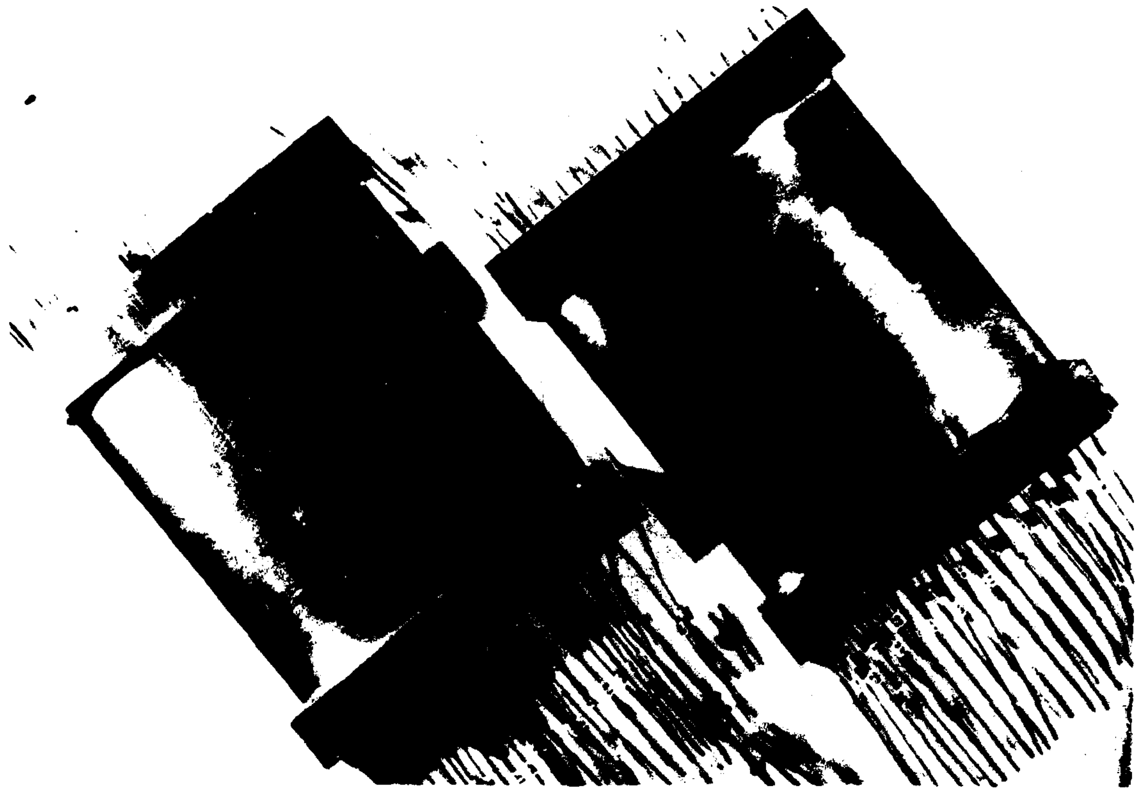


Figure 3. Photograph of the Pressure Models of the Two Profiles

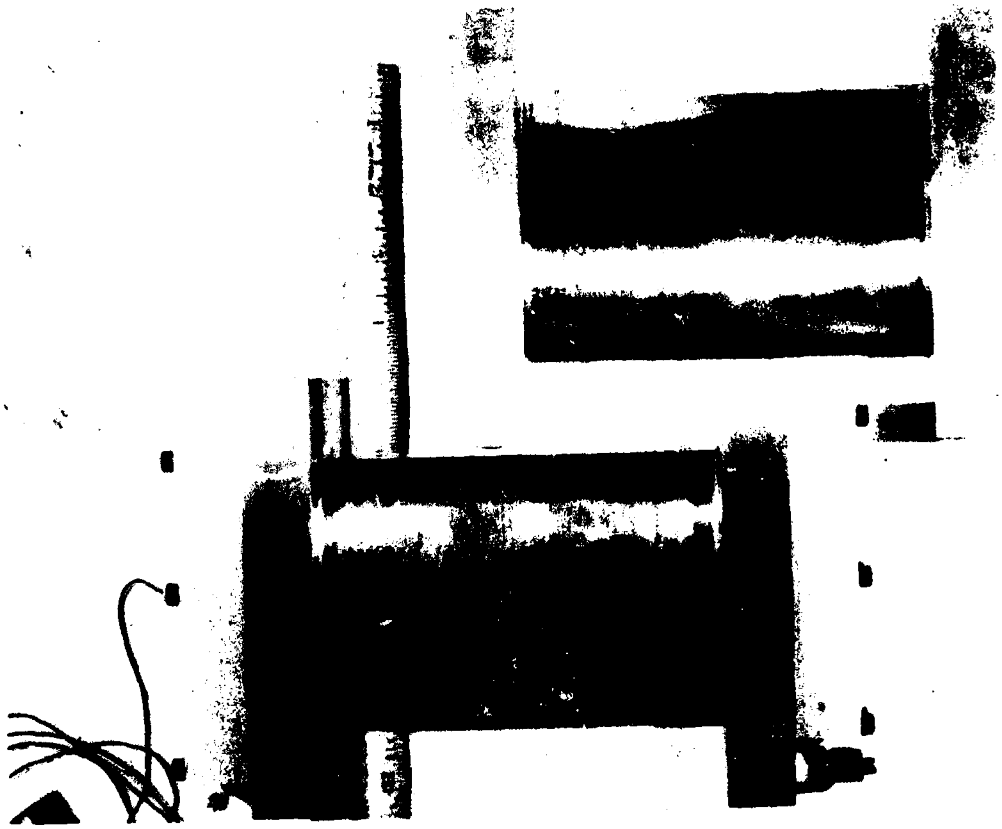
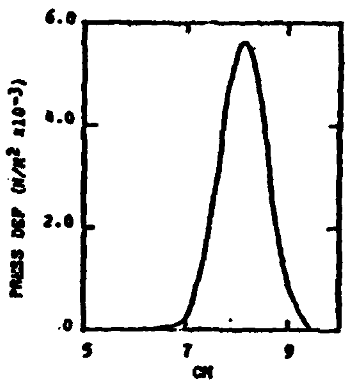
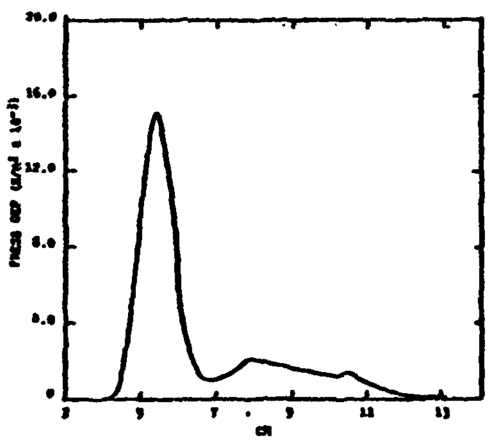


Figure 4. Photograph of the Thermal Models Used in the Study



(a) Subcritical Flow Over the Airfoil.



(b) Supercritical Flow Over the Airfoil.

Figure 5. Typical Pitot Pressure Profiles Obtained in the Wakes of Models in the Airfoil Tunnel.

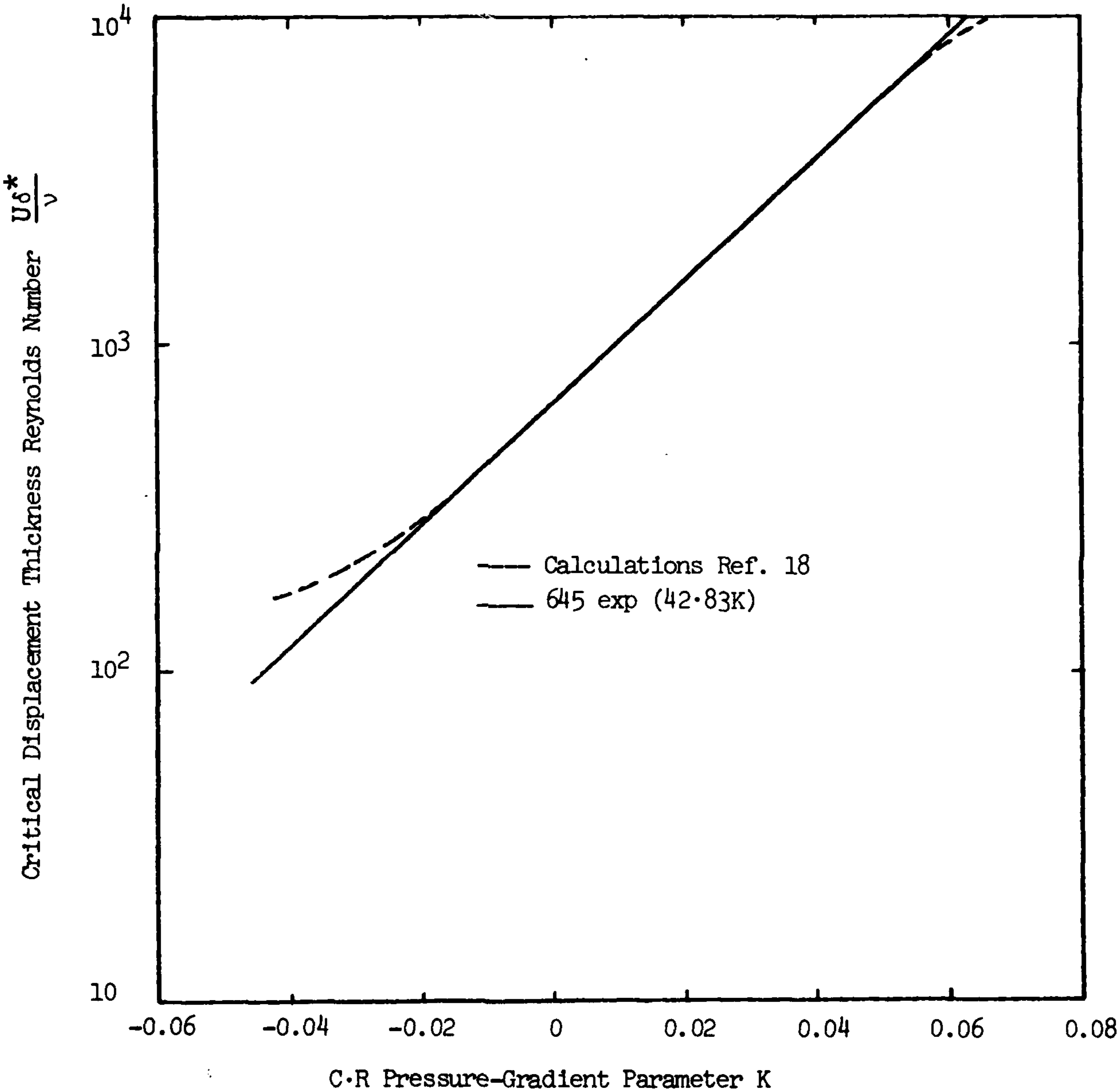


Figure 6. Comparison of Exponential Curve-Fit with Calculations from Stability Theory for Influence of Pressure-Gradient.

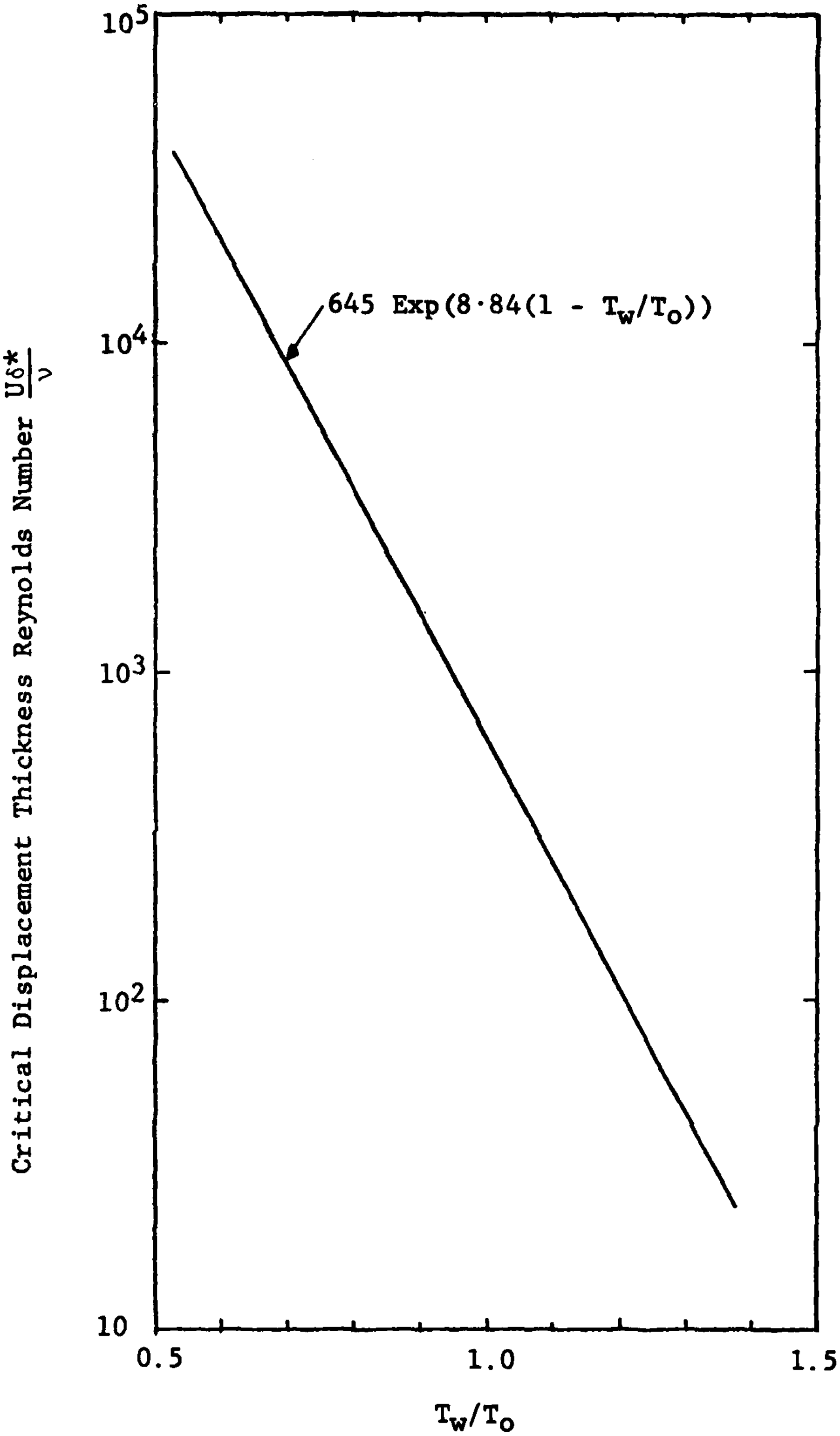


Figure 7. Best Exponential Fit to Calculations of Stability Theories for Heat Transfer on Flat Plate; Refs. 17 and 10.

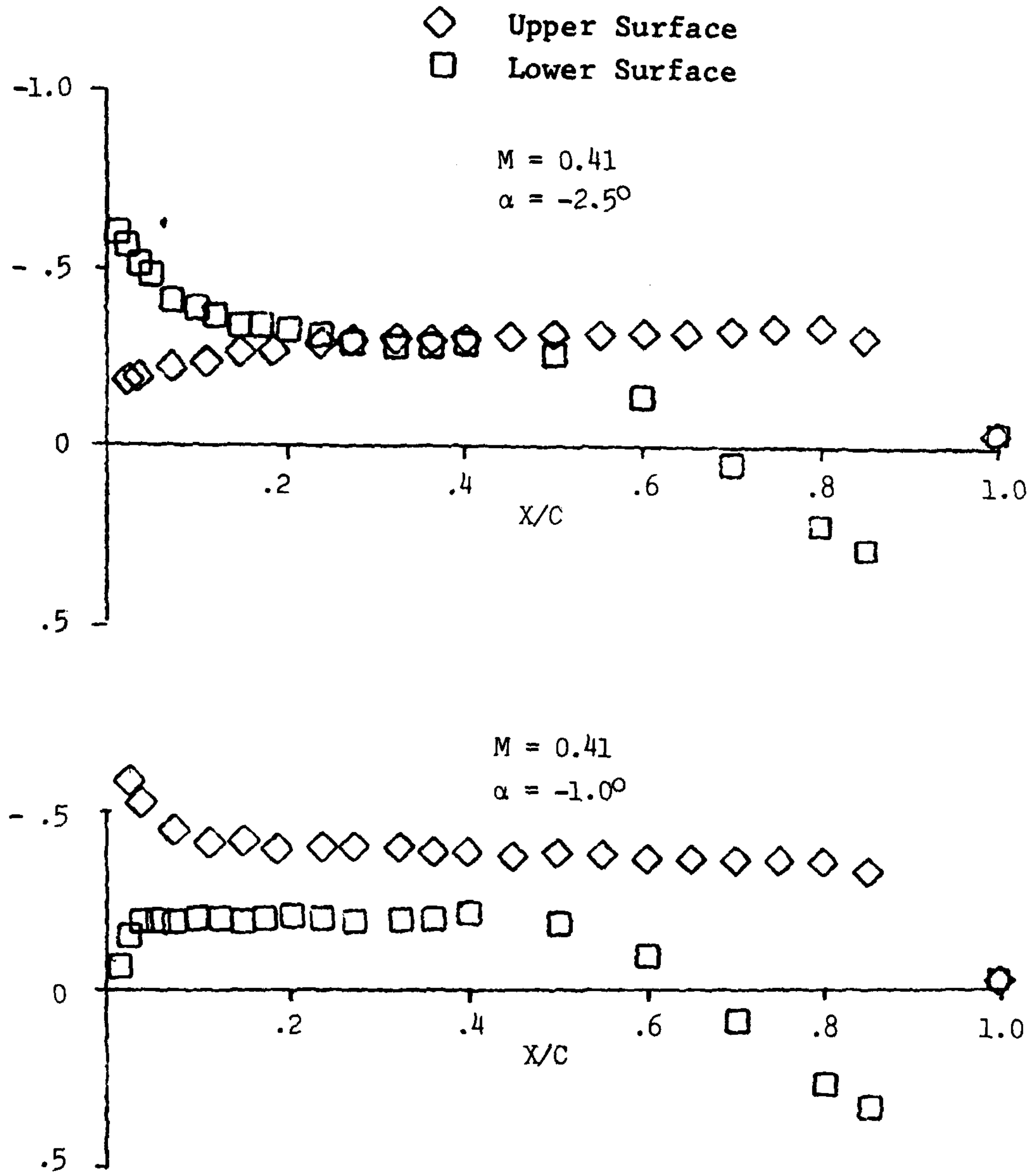


Figure 8. Pressure Distributions from Pressure Model of Aft-Loaded FDLA Profile at Attack Angles on Either Side of Test Condition.

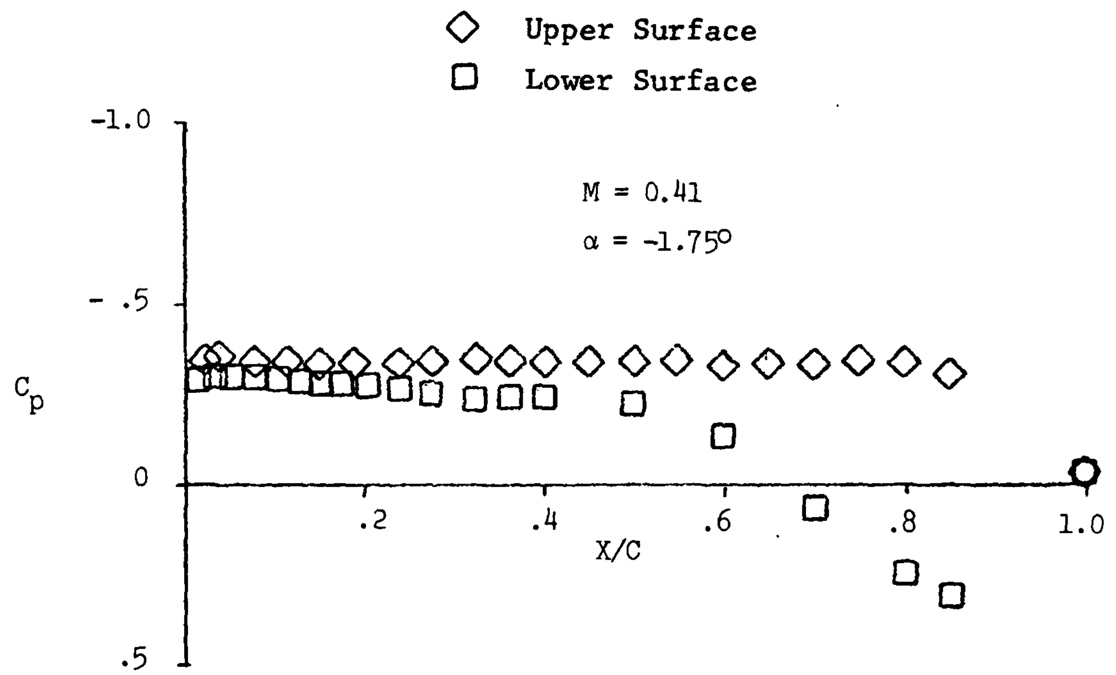


Figure 9. Pressure Distribution on Aft-Loaded FDLA Profile at Selected Test Condition.

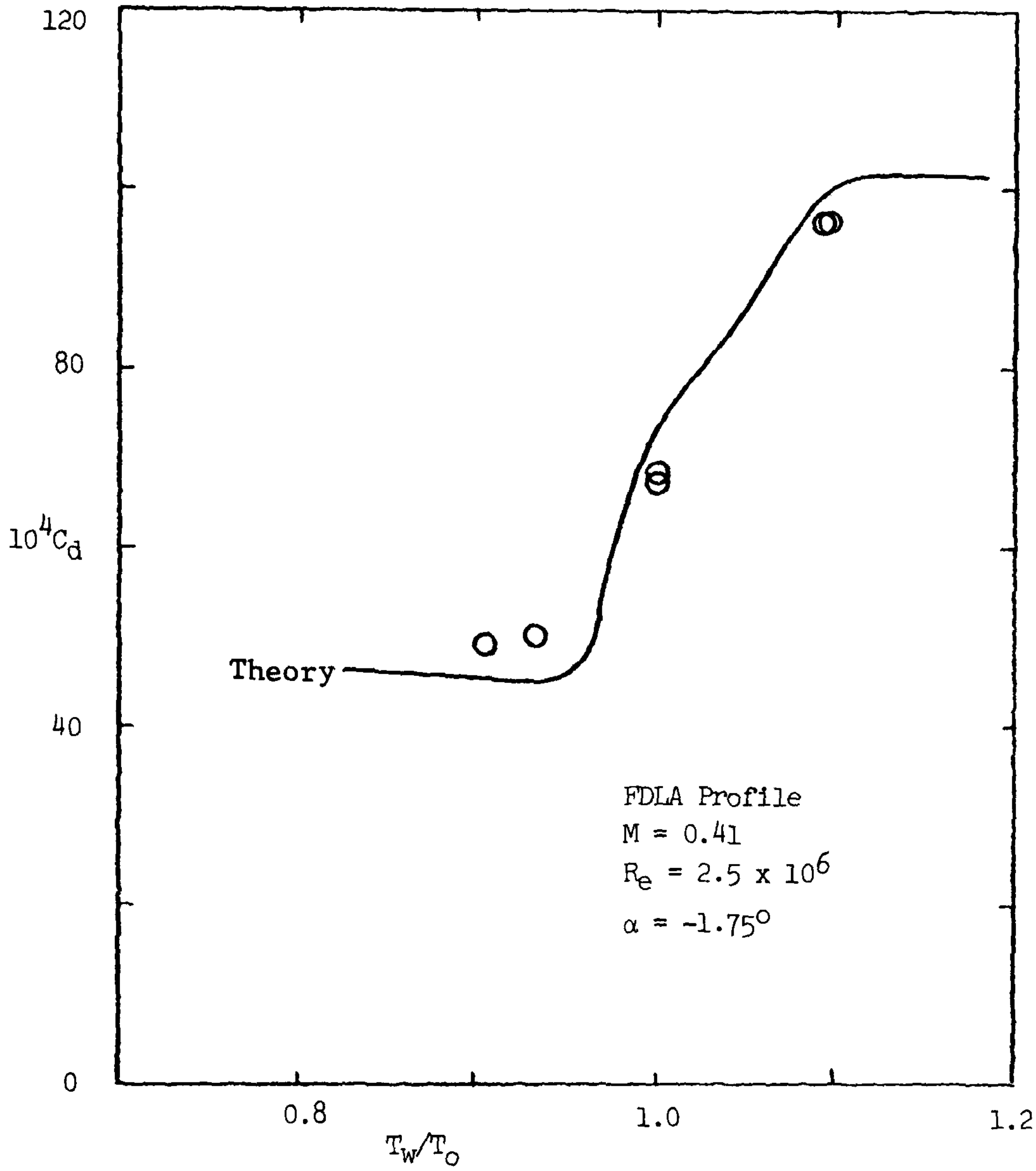


Figure 10. Variation in Drag Coefficient of Aft-Loaded FDLA Profile with Surface-to-Stagnation Temperature Ratio for Condition of Figure 9.

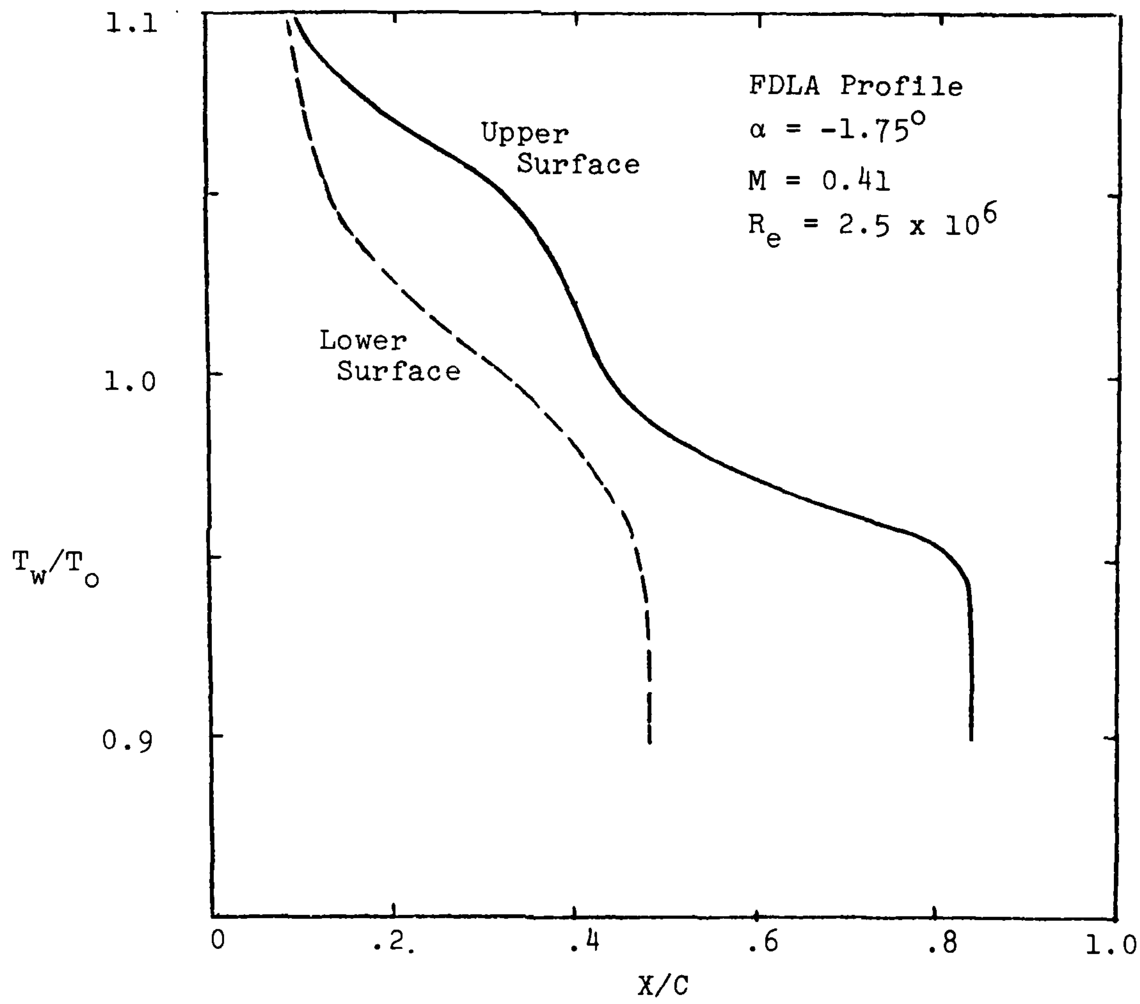


Figure 11. Theoretical Prediction of Transition Location on Aft-Loaded Airfoil for Condition of Figure 9

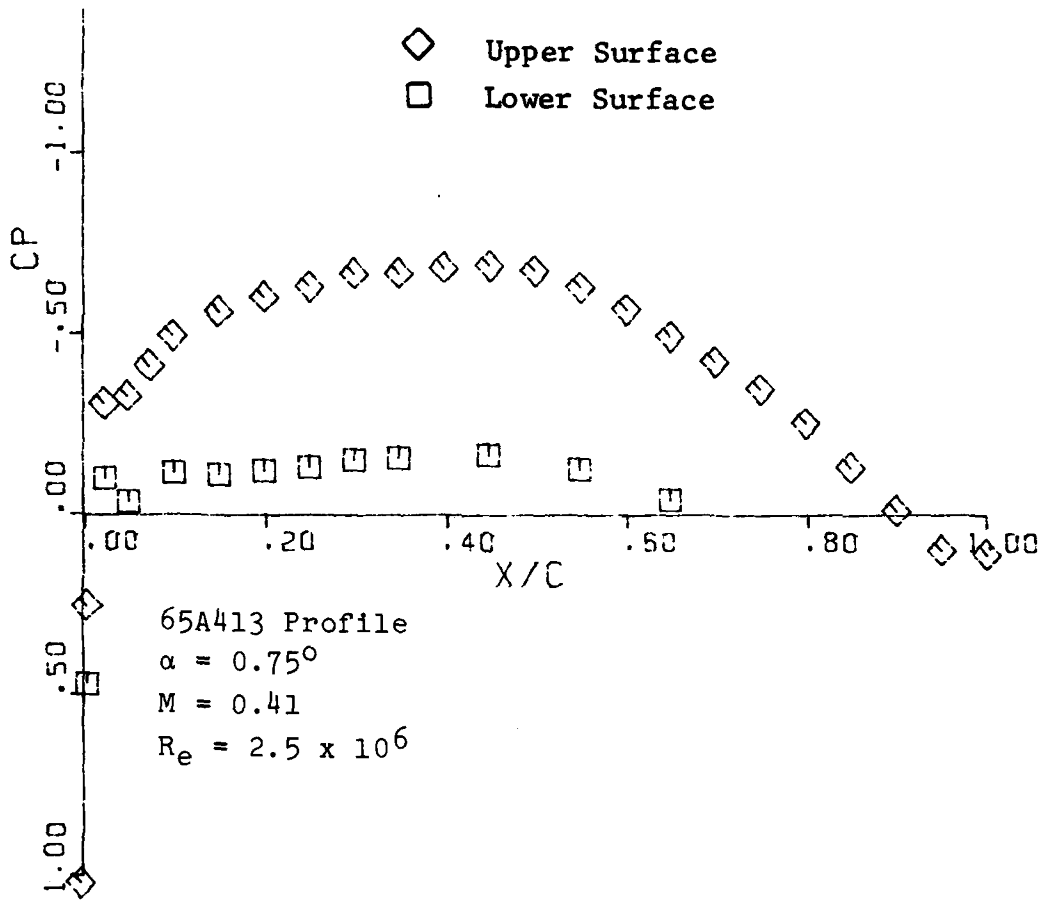


Figure 12. Pressure Distribution on the NACA 65A413 Profile of the Selected Test Condition.

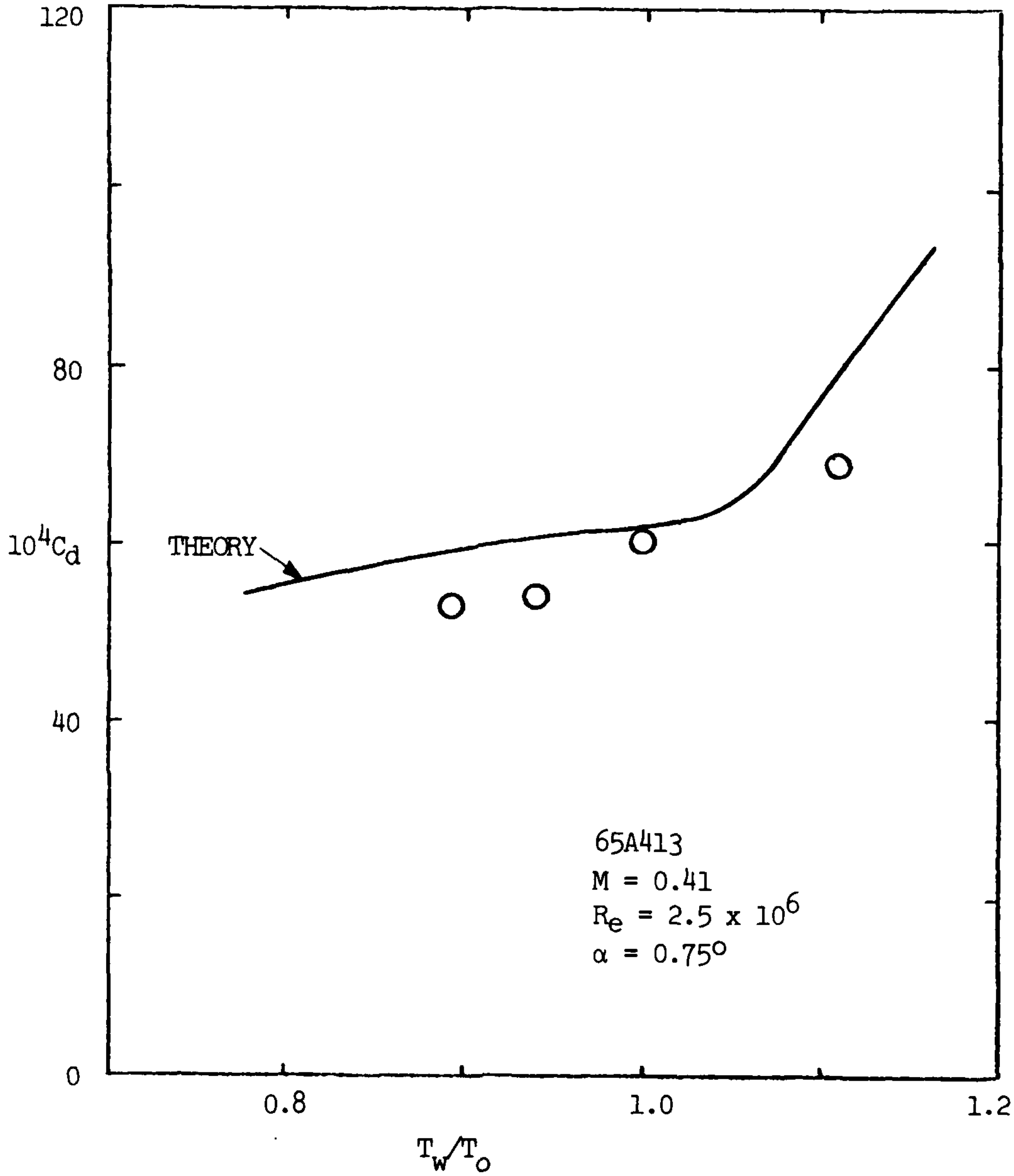


Figure 13. Variation in Drag Coefficient of NACA 65A413 Profile with Surface-to-Stagnation Temperature Ratio for Condition of Figure 12.

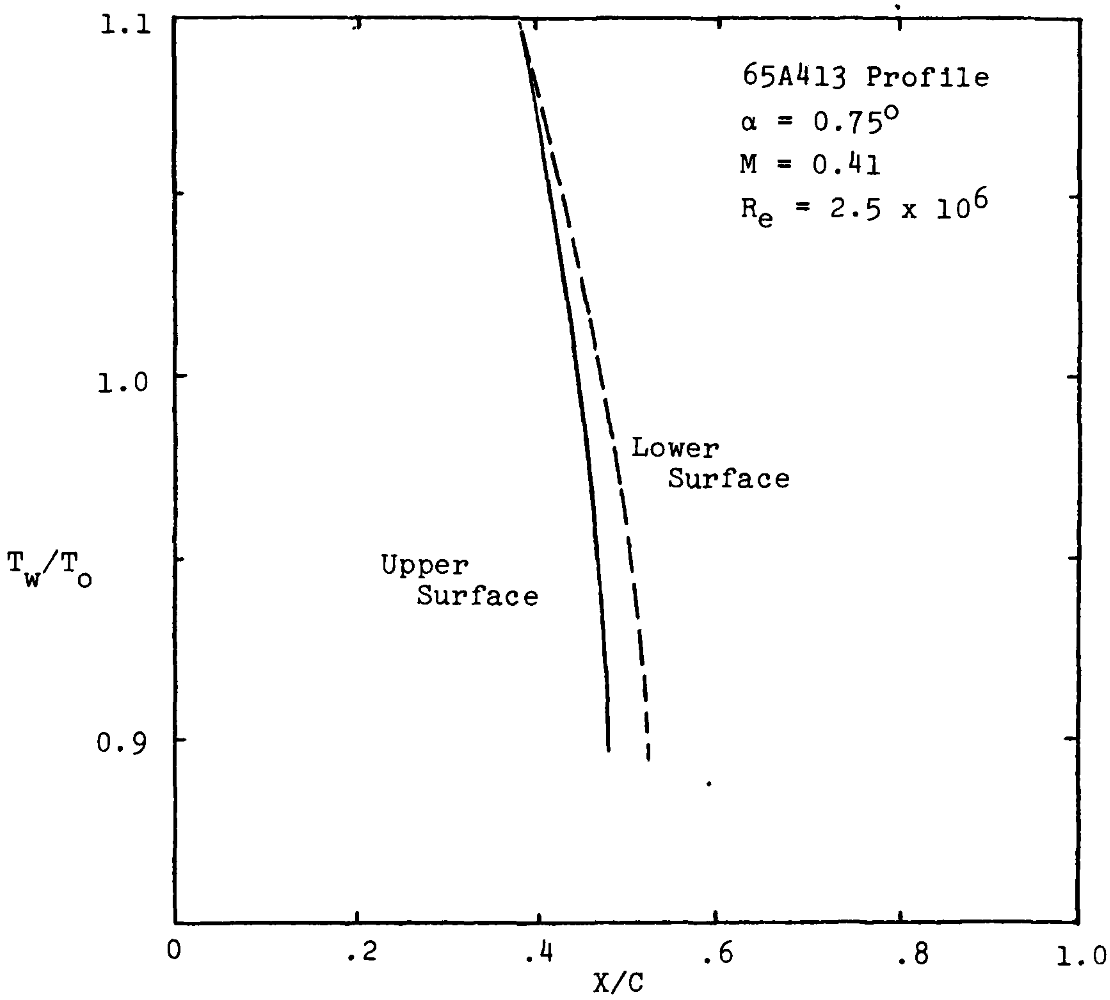


Figure 14. Theoretical Prediction of Transition Location on NACA 65A413 Airfoil for Condition of Figure 12.

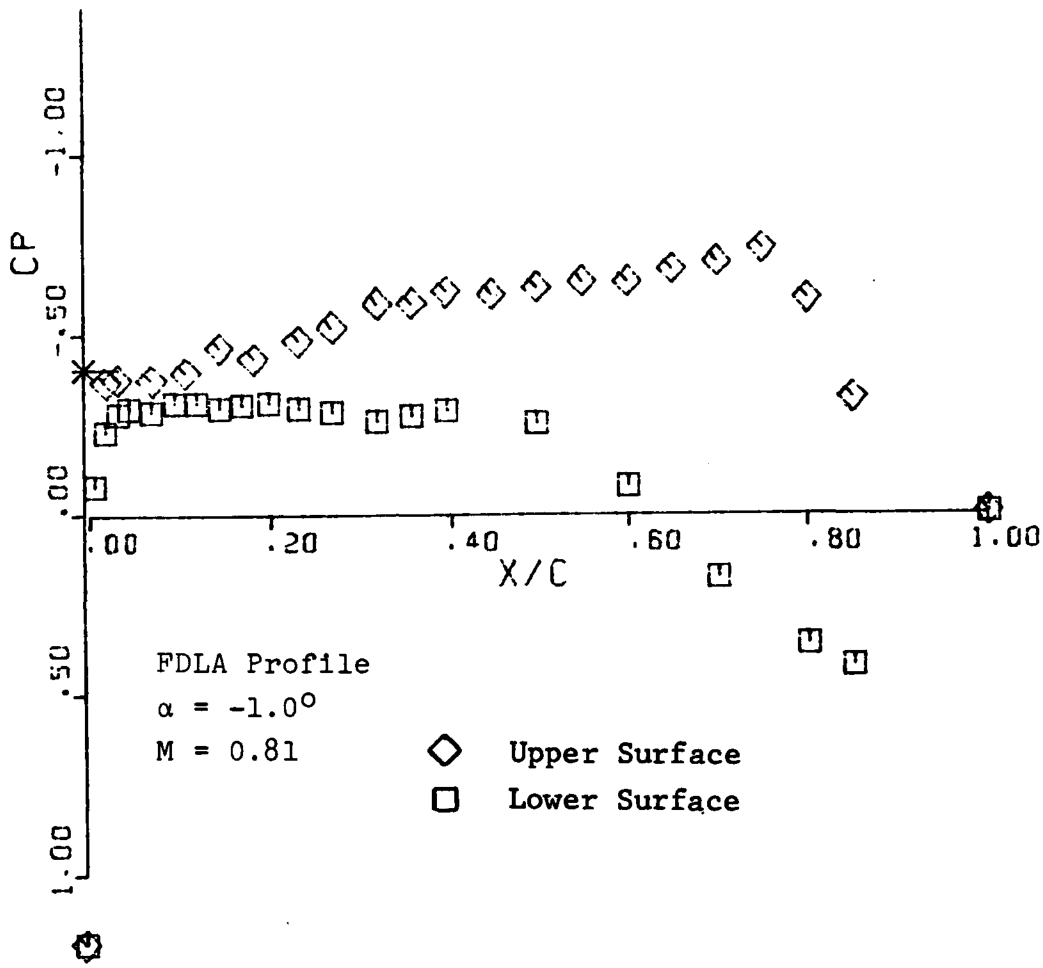


Figure 15. Pressure Distribution on the Aft-Loaded Airfoil at the Selected Test Condition for Supercritical Flow.

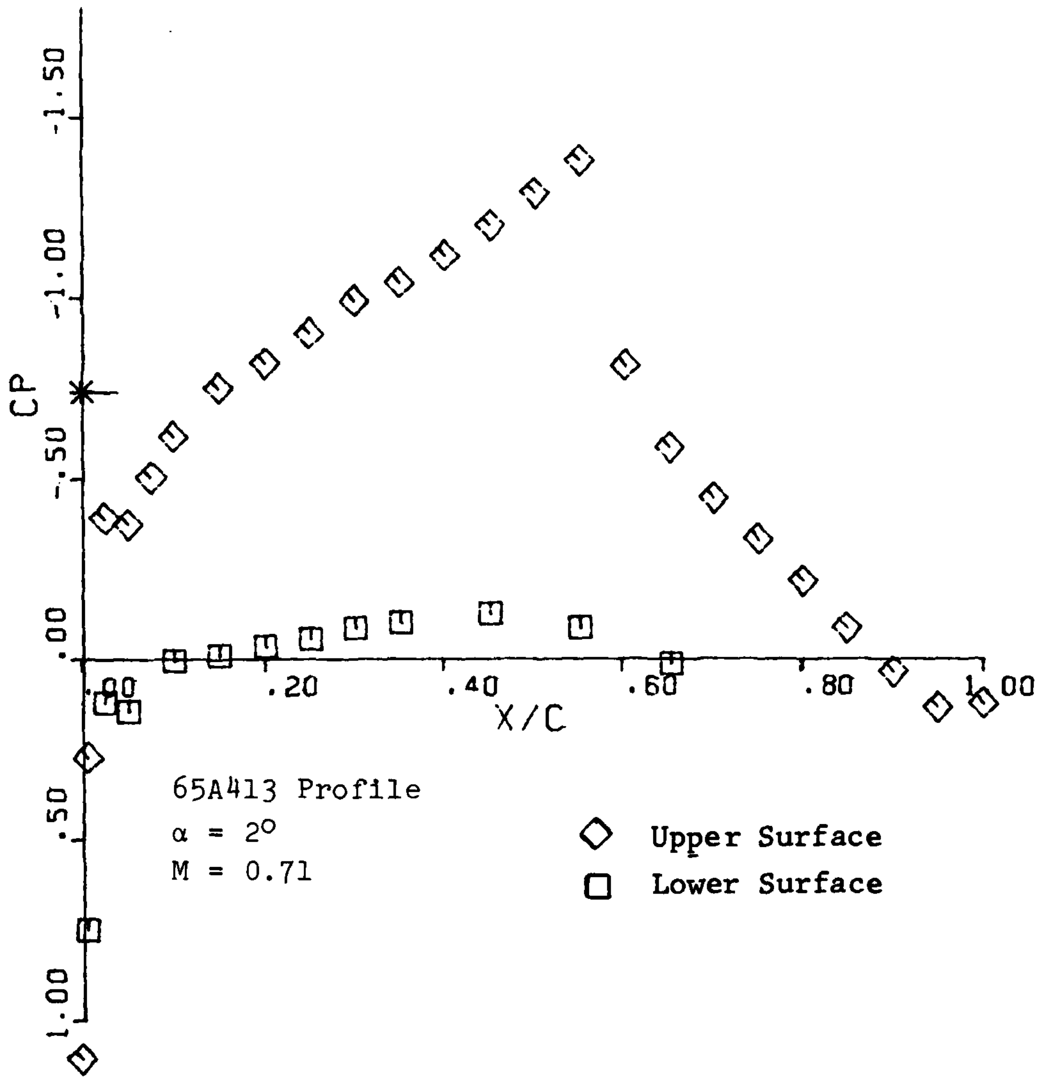


Figure 16. Pressure Distribution on the NACA 65A413 Airfoil at the Selected Test Condition, Supercritical Flow

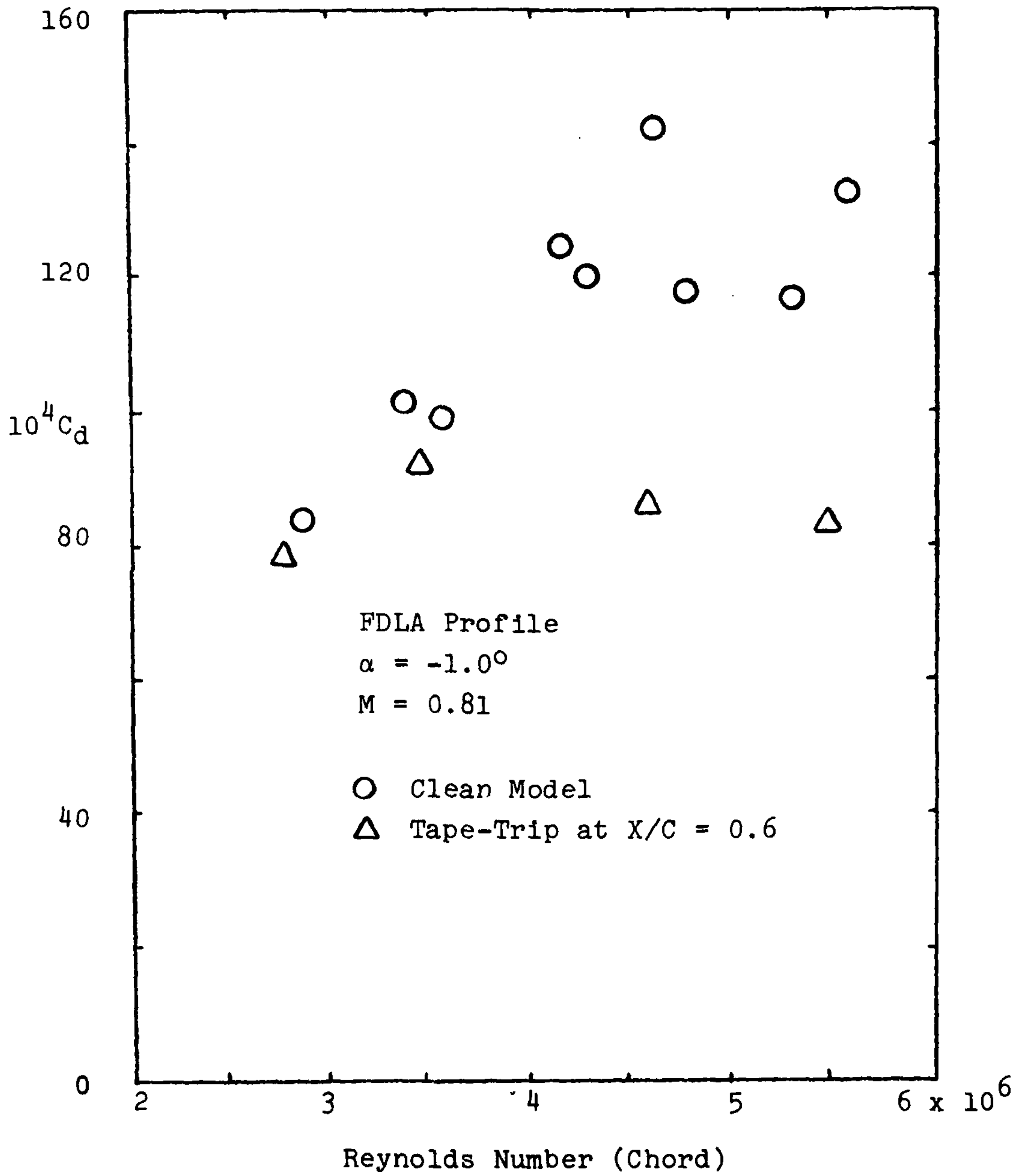


Figure 17. Drag Measurements for the Aft-Loaded Airfoil at the Condition of Figure 15 Showing Influence of Trip Ahead of Separation Point.

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